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THE EFFECTS OF SYSTEMATIC VARIATION OF SEVERAL SHAPE  
PARAMETERS ON THE CHARACTERISTICS OF AIRFOIL  
SECTIONS AT HIGH-SUBSONIC MACH NUMBERS

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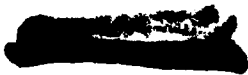
INTRODUCTION

The need for additional information on the characteristics of thin airfoil sections at high subsonic Mach numbers is apparent to all those actively engaged in the design of airplane lifting surfaces for transonic Mach number applications. In the summer of 1948, a systematic program of wind-tunnel investigations to provide some of the desired information was formulated jointly by the NACA and the aircraft industry through their representatives on the NACA Special Subcommittee on Research Problems of Transonic Aircraft Design. The principal objective of this program was the assessment of the effects on the characteristics of thin airfoil sections of systematic variations of trailing-edge angle, leading-edge radius, camber, thickness distribution, and thickness-chord ratio at Mach numbers approaching unity. The purpose of this paper is to summarize briefly the results of the experimental investigations.

Most of the data have been obtained from tests of 6-inch-chord airfoils in the Ames 1- by  $3\frac{1}{2}$ -foot high-speed tunnel at Mach numbers from 0.3 to a maximum of 0.92 and at Reynolds numbers which varied correspondingly from approximately 1 to  $2 \times 10^6$ .

TRAILING-EDGE ANGLE

There has been much speculation concerning the influence of the trailing-edge angle on the characteristics of airfoil sections at high subsonic Mach numbers, but to date there has been little real information of a systematic nature on the subject. The stimulus for interest in this geometric parameter consists chiefly in reports of poor lift-curve slopes and control-surface-effectiveness characteristics associated with trailing-edge angles greater than  $18^\circ$  (references 1 to 4). In an effort to isolate the effect of this variable and to provide a basis for a more detailed study of the problem, a preliminary



experimental investigation was undertaken in the Ames 1- by  $3\frac{1}{2}$ -foot high-speed wind tunnel.

The aerodynamic characteristics of a 10-percent-chord-thick airfoil section, both alone and with a 25-percent-chord plain flap, were determined for trailing-edge angles of  $6^\circ$ ,  $12^\circ$ , and approximately  $18^\circ$ . The characteristics for the profile without a flap are reported in reference 5. The airfoil thickness distribution was chosen as that of the modified NACA four-digit series. (See reference 6.) This thickness distribution is expressed by a fourth-power equation which permits the trailing-edge portion of the profile to be varied without essentially changing the shape forward of the maximum thickness position. The trailing-edge shapes investigated are illustrated in figure 1.

The only appreciable effects of the trailing-edge-angle variation on the characteristics of the airfoil without a flap were observed in the lift-curve slope, maximum lift-coefficient variation with Mach number, and the drag-divergence Mach number at low lift coefficients. In figure 2, the lift-curve slope  $dc_l/d\alpha$  at  $0^\circ$  angle of attack is shown as a function of Mach number  $M$  for the three trailing-edge angles. The differences are small and of no particular importance in that the variation with Mach number was not significantly changed. This result is nothing like that of G6thert in reference 1, where a pronounced effect of trailing-edge angle was observed on the lift-curve slope of a 15-percent-chord-thick airfoil section. This would seem to indicate a lessening influence of trailing-edge angle with decreasing thickness-chord ratio.

In figure 3, an improvement in the maximum section lift coefficient  $C_{l_{max}}$  at Mach numbers above about 0.7 is seen to accompany a reduction in the trailing-edge angle. Reduction of the trailing-edge angle adversely affected the drag-divergence Mach number  $M_d$  of the airfoil section at lift coefficients near zero as is evidenced in figure 4. The difference at zero lift coefficient over the range of angles investigated amounted to approximately 0.04 Mach number. At lift coefficients above 0.2, the difference disappeared.

The effects of changes in trailing-edge angle on the variation with Mach number of the lift effectiveness of a plain flap are shown in figure 5. In this figure, the rate of change of section lift coefficient with flap deflection  $dc_l/d\delta$  for deflections from  $-2^\circ$  to  $6^\circ$  is shown as a function of Mach number for the three trailing-edge angles and for angles of attack of  $0^\circ$ ,  $4^\circ$ , and  $6^\circ$ . In the zero-lift case, an abrupt loss of effectiveness beginning at a Mach number in the vicinity of 0.8 is evident for all trailing-edge angles. The interesting feature of these results is the very small benefit derived from

reduction of the trailing-edge angle even to a value as low as  $6^\circ$ . The only favorable effect of the decrease in trailing-edge angle was the elimination of the reversal of effectiveness indicated for the  $18^\circ$  angle. This result is not too surprising because, from visual observations of the flow field at zero angle of attack and small flap angles, the flap lay entirely within the region of separated flow aft of the compression shock on the airfoil and therefore could develop virtually no lifting pressures.

At the higher angles of attack, reduction of the trailing-edge angle did effect an improvement in the variation of the flap effectiveness with Mach number. It is probable that, had the investigation been extended to encompass larger flap deflections, the beneficial effects of trailing-edge-angle reduction would have been noted even for the lower angles of attack.

From the results of this and free-flight investigations (references 7 and 8), it is fairly obvious that the trailing-edge angle alone is not the governing airfoil-shape parameter in the variation of control-surface effectiveness with Mach number. Satisfactory effectiveness cannot be assured at all lift coefficients merely by holding the trailing-edge angle to a value less than, say,  $10^\circ$  or  $12^\circ$ , which has been tacitly accepted in some quarters as an upper limit for satisfactory characteristics.

#### LEADING-EDGE RADIUS

An analysis of the characteristics at high Mach numbers of a large number of airfoil sections has indicated the shape of the forward portion of an airfoil to be an important parameter governing these characteristics. To a first order this shape is expressed by the leading-edge radius. In the course of a preliminary investigation (reference 5) of the influence of this parameter, the characteristics of a 10-percent-chord-thick airfoil of the modified NACA four-digit series have been determined for leading-edge radii of 1.10, 0.70, and 0.27 percent of the airfoil chord. The nose shapes investigated are illustrated in figure 6. The leading-edge-radius variation was accomplished without altering the profile aft of the maximum thickness position.

The effect of the variation in leading-edge radius on the lift-curve slope of the airfoil section is shown in figure 7 to be unimportant. Figure 8 demonstrates a small favorable effect of reduction in leading-edge radius on the maximum section lift coefficient at Mach numbers above 0.65. The results shown in figure 9 indicate that Mach numbers of drag divergence were decreased somewhat at low lift

coefficients with decreasing radius. The effects on all these characteristics were considerably smaller than those noted previously for the variation in trailing-edge angle. The pitching-moment characteristics were not significantly affected by the changes in leading-edge radius.

The effects of similar variations of leading-edge radius on 4- and 6-percent-chord-thick sections were not sufficiently important to warrant discussion.

### CAMBER

The effects of large camber variation on the characteristics of a 10-percent-chord-thick airfoil section at high Mach numbers have recently been determined from tests of an NACA 64A-series profile cambered for design lift coefficients ranging from 0 to 0.9. In figure 10, the lift-divergence Mach number  $M_l$  is plotted as a function of lift coefficient for the various design lift coefficients  $c_{l1}$ . It is obvious that, for applications calling for operating lift coefficients up to 0.5, the symmetrical section would be the most desirable. For lift coefficients greater than 1.0, the sections cambered for design lift coefficients of 0.6 and 0.9 would afford the best characteristics. Similarly, in figure 11, the value of camber in providing a larger range of lift coefficient for favorable drag-divergence characteristics is indicated.

The variation of maximum lift coefficient with Mach number for the various amounts of camber is illustrated in figure 12. At Mach numbers below about 0.6, by virtue of the relatively low test Reynolds numbers, the results cannot be used with assurance in the prediction of large-scale characteristics. Ample evidence exists (reference 9), however, to indicate that at the higher Mach numbers, the influence of Reynolds number on the maximum lift coefficient is small. It is interesting to note that the beneficial effect of camber on the maximum lift coefficient persists throughout the Mach number range of the investigation.

In figures 13 and 14, respectively, are shown the variations with Mach number of the angle of attack for lift coefficients of 0 and 0.9 for the various amounts of camber. The familiar adverse effects of camber on the longitudinal trim characteristics of straight-wing airplanes employing such wing sections are evident here. The variations of angle of attack for intermediate lift coefficients lie within those shown on these two figures. The variation of lift-curve slope with Mach number at the design lift coefficient is shown in figure 15 for each of the cambered sections. It is this unfavorable effect of camber on the lift-curve slope coupled with the previously indicated adverse

lift characteristics for airplane trim (figs. 13 and 14) which makes the cambered sections inferior to the symmetrical profiles for high-speed straight-wing airplanes.

In the case of swept wings, however, the position of camber should be reappraised. The theoretical foundations upon which two-dimensional airfoil data may safely be applied to the design of swept wings are yet to be laid; but sufficient evidence has been obtained to indicate the usefulness of section characteristics in such cases if the stream velocity be considered resolved into components normal and parallel to what might be termed the lifting axes of the wing and the section be considered as that normal to such axes. The lift characteristics of thin symmetrical sections handicap the performance of swept-wing airplanes in both the landing and high-altitude, high-speed flight conditions. Utilization of large amounts of camber in the sections comprising such wings therefore becomes desirable. Furthermore, for swept wings, it is possible that, if the high positive camber desirable for landing and high-speed high-altitude performance is suitably distributed along the span of the wing, the trim changes promoted by the camber will give to an airplane in an overspeed condition a nosing-up tendency in place of the diving tendency noted for the straight-wing airplane. That is, for highly cambered wing tip sections, the lift carried at the tips will be lost (as the lift-divergence Mach numbers of these sections are exceeded) before that of the lower cambered inboard sections and, by virtue of the large longitudinal moment arm of the tip region, a nosing-up moment will be experienced by the airplane. If the nose-up is not too rapid, this characteristic might even be considered a favorable one for a bomber-type airplane.

#### THICKNESS DISTRIBUTION

Some of the principal questions that have been raised concerning the effects of thickness distribution on section characteristics of thin airfoils are: (a) what is the effect of removing the cusp from the trailing edge of a low-drag airfoil, (b) how do the characteristics of the NACA four-digit-series (conventional) airfoils compare with those of the NACA six-series (low drag) family, and (c) how does changing the position of maximum thickness affect the properties of conventional airfoils? In order to answer these questions, section data were procured for four 10-percent-thick airfoils considered sufficiently representative to permit generalization of the results. The airfoils chosen were the NACA 64-010, 64A010, 0010, and 0010-64. The characteristics of the first two airfoils are reported in reference 10, and those of the latter two in reference 11.

Curves summarizing the lift characteristics are presented in figures 16 to 18. Figure 16 illustrates the variation of lift-curve slope with Mach number. It is immediately apparent that this parameter is unaffected by the presence or absence of a cusped after-profile by a change in the position of maximum thickness from 30 to 40 percent of the chord for the conventional sections, or even by the differences in profile between low-drag and conventional sections. Similar observation can be made with respect to the Mach number of lift divergence (fig. 17). The maximum lift coefficients, however, shown in figure 18, are considerably greater at Mach numbers above 0.7 for the low-drag than for the conventional sections.

The Mach number for drag divergence (fig. 19) has been selected to illustrate the effects of removing the cusp from the low-drag airfoil, of a change in thickness distribution from that of a low-drag to that of a conventional section, and of shifting the maximum thickness position of a conventional section rearward. Although it is apparent that the absence of the cusp has no important effect on the Mach number for drag divergence for 10-percent-thick low-drag airfoils, one may conclude that the uncambered low-drag airfoils are superior in this respect to conventional sections at lift coefficients above 0.4; and also that, for conventional airfoil sections at low lift coefficients, a considerable gain may accrue from shifting the maximum-thickness location rearward.

#### THICKNESS-CHORD RATIO

Ample evidence has been obtained (references 12 and 13) to indicate the favorable effect of reduction in thickness-chord ratio  $t/c$  upon the characteristics of airfoil sections at high Mach numbers. No information has been available, however, on the effects of a systematic reduction of thickness-chord ratio down to 4 percent for a single thickness form. The results of a recently completed investigation of the characteristics of four symmetrical NACA four-digit-series airfoil sections ranging in thickness from 10 to 4 percent of the chord therefore become of interest. The thickness distribution investigated was that of the NACA 00XX-64 family of profiles.

From the variation of lift-curve slope with Mach number shown in figure 20 for the various thickness-chord ratios, significant effects are apparent only at the higher Mach numbers and are what should be expected in that each successive reduction of thickness delays the Mach number at which the lift-curve slope breaks. The trend and magnitude of the differences are somewhat more clearly illustrated in figure 21 which is a plot of the Mach number of lift divergence as a

function of thickness-chord ratio for three lift coefficients. In this figure, it is seen that the increase of lift-divergence Mach number amounts to approximately 0.1 for a reduction in thickness from 10 to 4 percent of the airfoil chord and that this improvement is realized at lift coefficients at least as large as 0.6.

Reduction of maximum thickness below 10 percent of the chord also has beneficial effects on the maximum lift coefficient attainable at the higher subsonic Mach numbers. (See fig. 22.) The reduction in thickness is observed to result in marked improvement at Mach numbers above 0.75. The values obtained at Mach numbers below about 0.6 may possibly suffer from the effects of low scale.

The effect on airfoil drag characteristics of reducing the maximum thickness to values as low as 4 percent of the chord is illustrated by the variation of the Mach number for drag divergence with thickness-chord ratio for two different values of the lift coefficient. (See fig. 23.) For the sacrifice in thickness-chord ratio from 10 to 4 percent the gain in drag-divergence Mach number is relatively small. This result, however, is essentially that which would be predicted from consideration of the critical Mach number variation.

It may be stated, therefore, that, within the range of subsonic Mach numbers investigated, the effects on lift characteristics of reducing the maximum thickness-chord ratio are both large and beneficial. The corresponding effects on drag, although appreciable, may not be sufficiently great in themselves to justify the structural complexity required in the utilization of thickness-chord ratios as low as 4 percent for transonic aircraft.

#### SUMMARY AND CONCLUSIONS

In summary, the attempt has been made to give a general view of the effects of a systematic variation of five major geometric variables on the more important characteristics of thin airfoil sections at high subsonic Mach numbers. The principal conclusion drawn is that, save for the effect of trailing-edge angle on control-surface effectiveness, camber and maximum thickness are the only shape parameters which decisively influence the characteristics of airfoil sections of 10 percent and less thickness-chord ratio at these Mach numbers. Stated in another manner, given a profile of particular camber and a low thickness-chord ratio, the choice of values for the other shape parameters is of little consequence as far as the high-speed characteristics of the airfoil sections are concerned.

It follows from this reasoning that, in the choice of thickness distribution for an airfoil of 10 percent or less thickness-chord ratio, considerable freedom may be exercised to obtain a desirable characteristic at low speeds without compromising the characteristics at high speeds. The import of this conclusion is illustrated by the example to follow of what was accomplished in this respect in one instance having important significance in the design of swept wings.

It has been found very difficult to provide highly swept wings with adequate maximum lift at low speeds. Camber has been employed to overcome this difficulty but the airfoil sections used have been those thought favorable to the promotion of good performance at high speeds. The sections have accordingly been of the NACA 6-series type with maximum-lift characteristics at low speeds known to be poorer than those of the NACA four-digit series which are characterized by more bulbous nose shapes. It was therefore reasoned that, if it were possible to employ the desired camber on a section of the latter type without seriously penalizing the high-speed characteristics, the low-speed difficulties of the swept wing would be materially lessened.

The work of Nitzberg, Crandall, and Polentz in reference 14, indicated that an NACA 0010 profile cambered for a design lift coefficient of 0.3 with an NACA  $a = 1.0$  mean line had characteristics at high speeds which were at least as good in several respects as those of an NACA 64A-series profile of comparable thickness considered to be an optimum section for high Mach number applications. A test to establish the relative merit of the two sections with respect to maximum-lift characteristics at low speeds was therefore made in the Ames 7- by 10-foot tunnel at a Reynolds number of approximately  $5 \times 10^6$ . The results of this test, along with a further evaluation of the characteristics at higher Mach numbers, are presented in figures 24 to 28.

In figure 24, the ratio of maximum section lift coefficient for the NACA four-digit series airfoil to that of the NACA 64A310 section is plotted as a function of Mach number. A gain of approximately 15 percent in the value of the maximum lift coefficient at low speeds would apparently be derived from the use of the NACA four-digit series thickness distribution over that of the NACA 6-series. As would be expected, this gain was not obtained without some sacrifice at higher Mach numbers, but, for the application in mind, it substantially outweighs the loss. The effects on the characteristics of lift-curve slope (fig. 25), angle of attack for the design lift coefficient (fig. 26), and lift-divergence Mach number (fig. 27) at high Mach numbers are of even less importance. In the case of drag-divergence Mach number (fig. 28), the NACA four-digit series section is somewhat inferior to the NACA 6-series section at lift coefficients above 0.4. In the design of swept wings, however, it may often be preferable to accept this penalty in return for improved lift at low speeds.



The significance of the foregoing result can perhaps not be over-emphasized, for it indicates the existence of a field of investigation that may yield answers to some of the vexatious problems of transonic airplane design.

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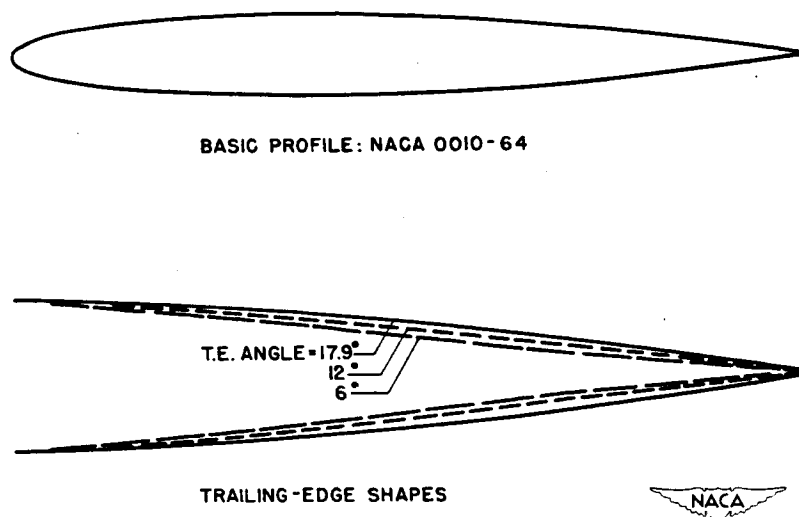


Figure 1.- Basic profile and trailing-edge shapes investigated.

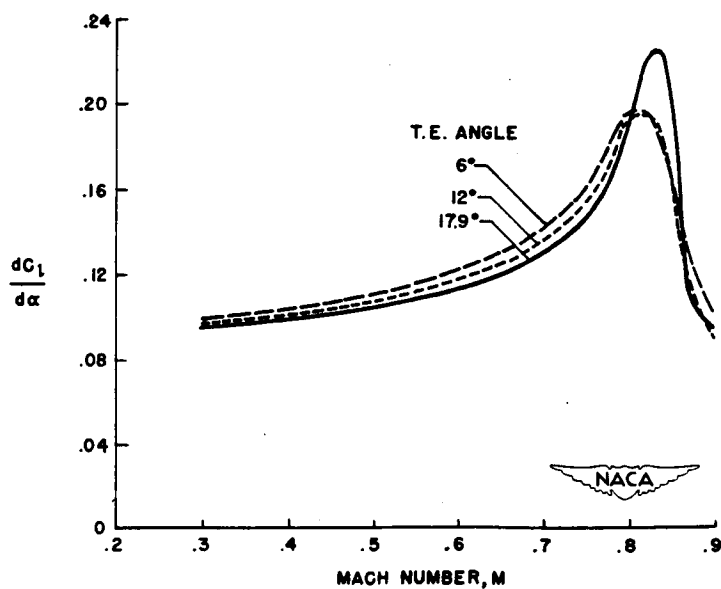


Figure 2.- Effect of trailing-edge angle on the variation of lift-curve slope with Mach number.

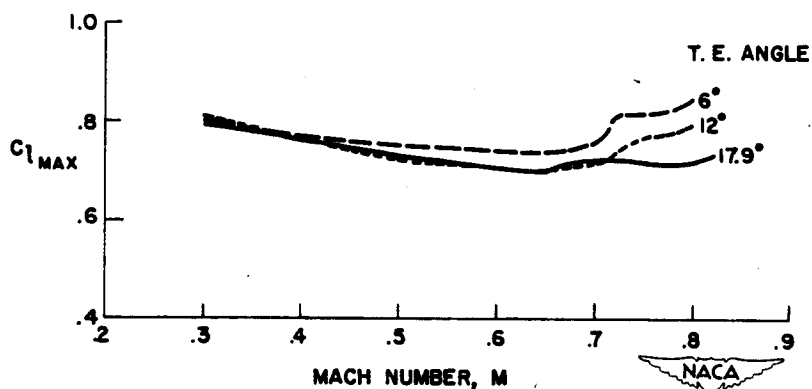


Figure 3.- Effect of trailing-edge angle on the variation of maximum lift coefficient with Mach number. ..

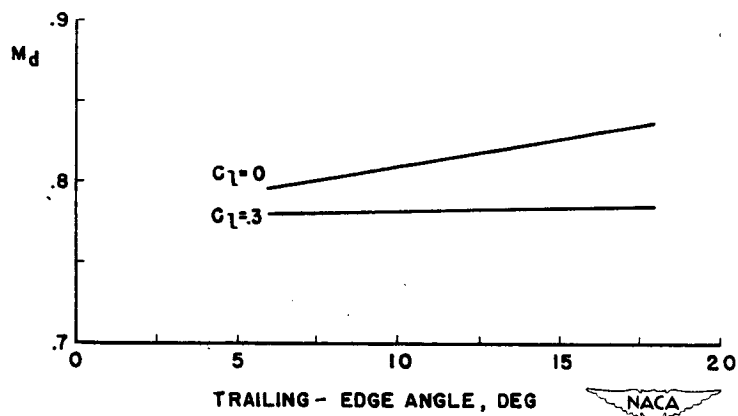


Figure 4.- Effect of trailing-edge angle on drag-divergence Mach number.

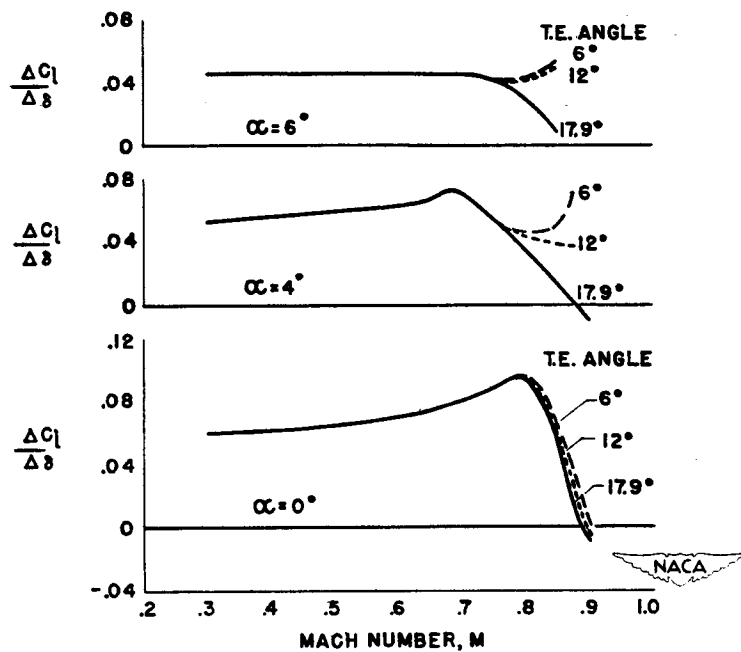


Figure 5.- Effect of trailing-edge angle on the variation of flap effectiveness with Mach number.

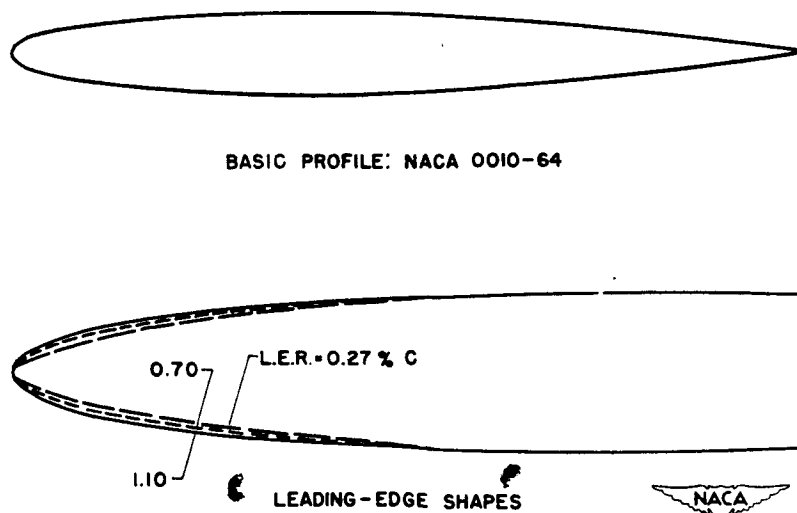


Figure 6.- Basic profile and nose shapes investigated.

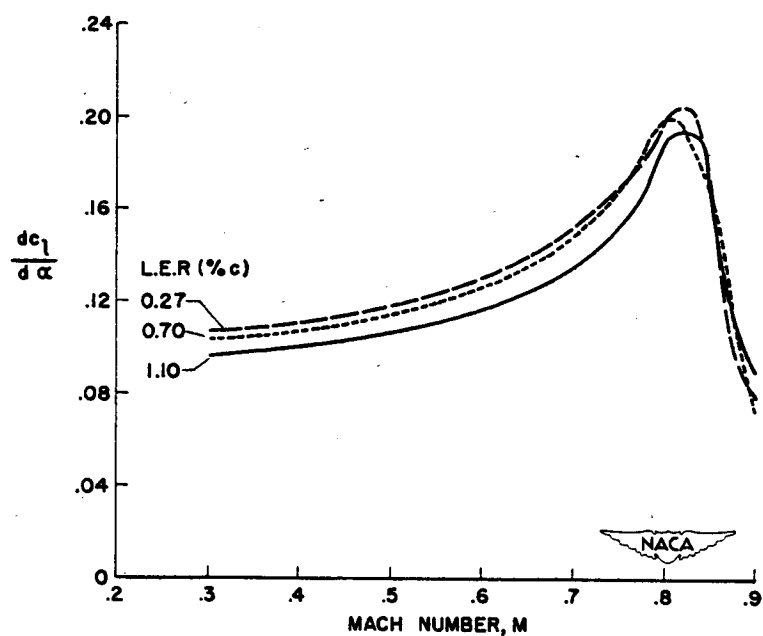


Figure 7.— Effect of leading-edge radius on the variation of lift-curve slope with Mach number.

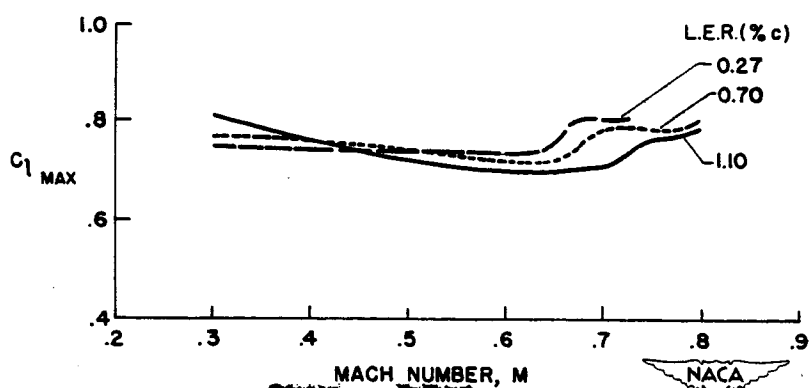


Figure 8.— Effect of leading-edge radius on the variation of maximum lift coefficient with Mach number.

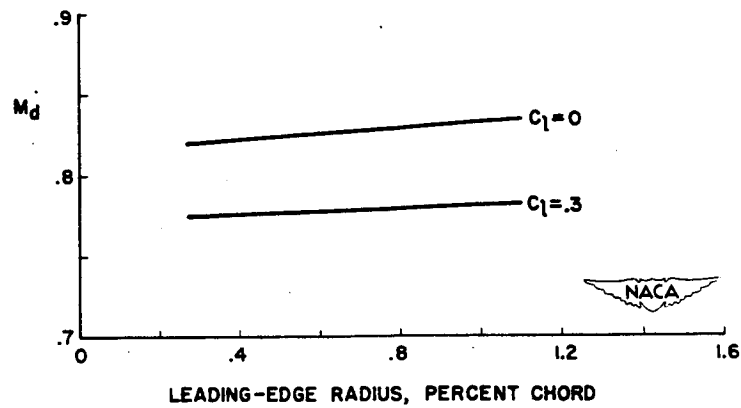


Figure 9.— Effect of leading-edge radius on drag-divergence Mach number.

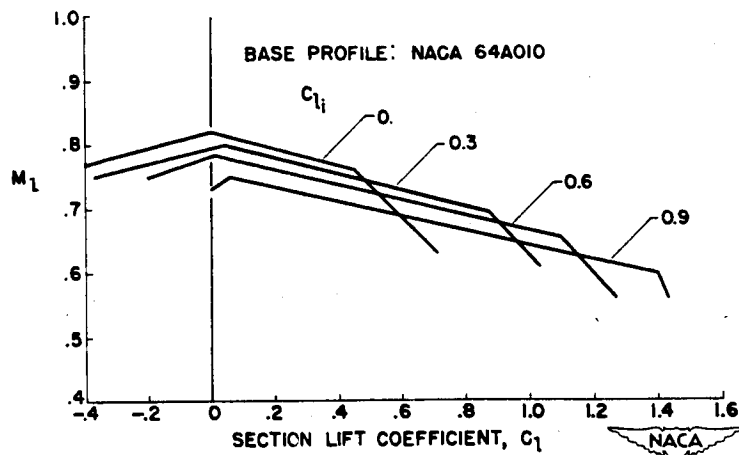


Figure 10.— Effect of camber on the variation of lift-divergence Mach number with lift coefficient.



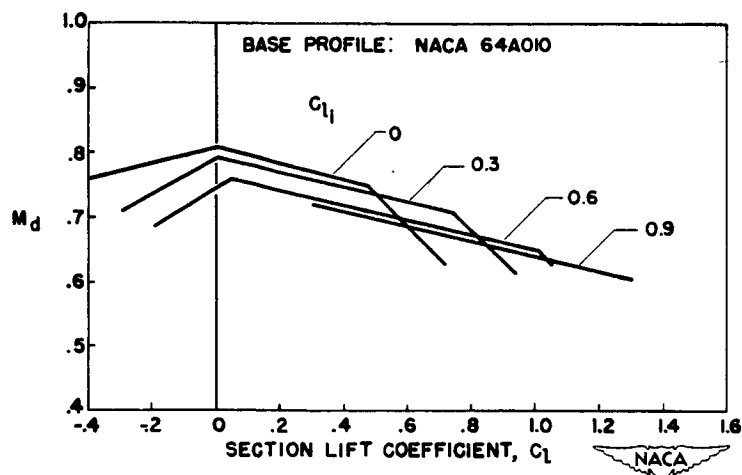


Figure 11.— Effect of camber on the variation of drag-divergence Mach number with lift coefficient.

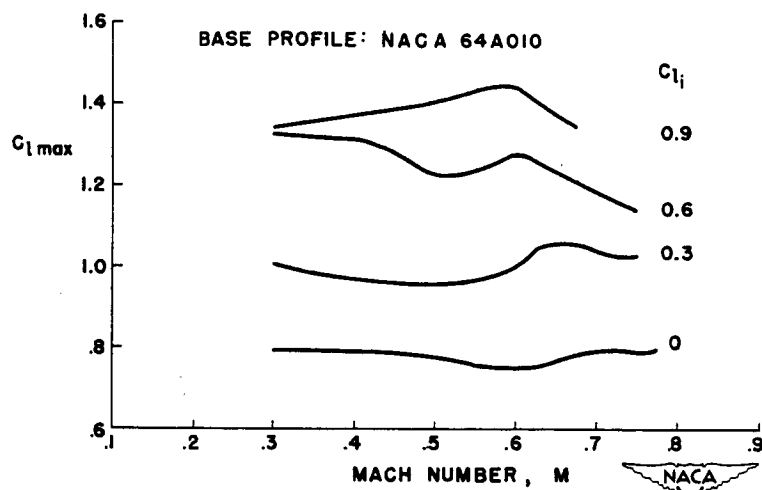


Figure 12.— Effect of camber on the variation of maximum lift coefficient with Mach number.

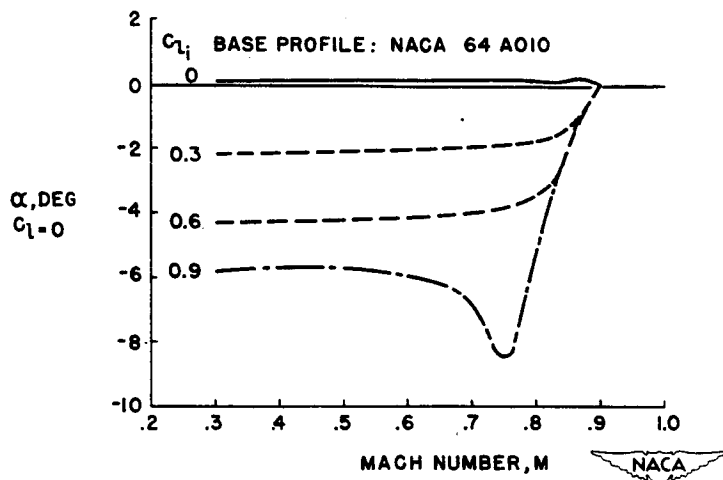


Figure 13.— Effect of camber on the variation with Mach number of the angle of attack for zero lift.

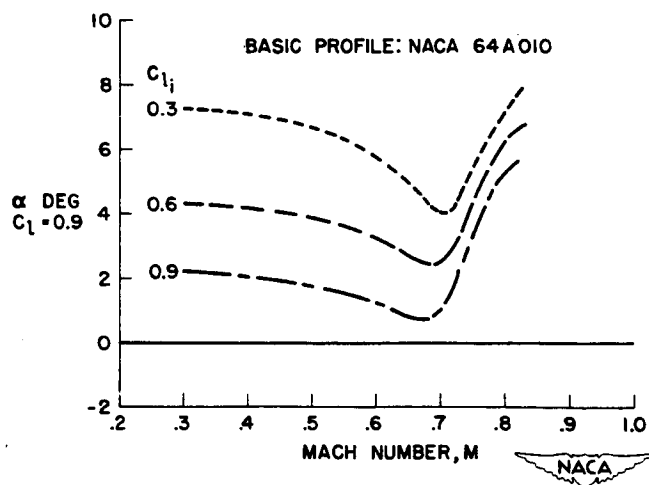


Figure 14.— Effect of camber on the variation with Mach number of the angle of attack for 0.9 lift coefficient.

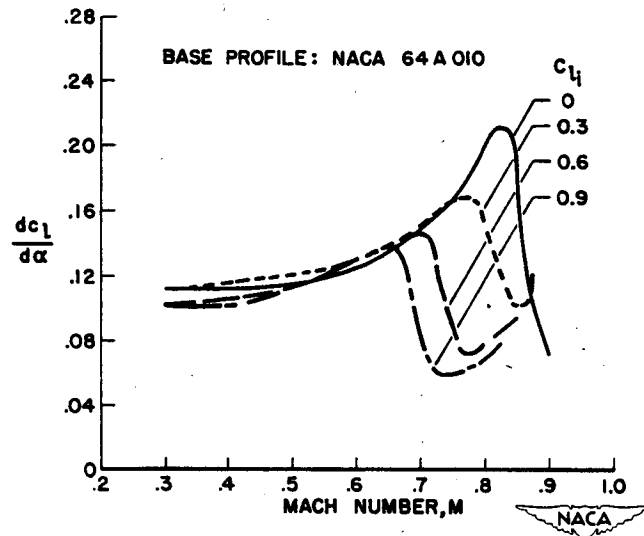


Figure 15.— Effect of camber on the variation with Mach number of the lift-curve slope at the design lift coefficient.

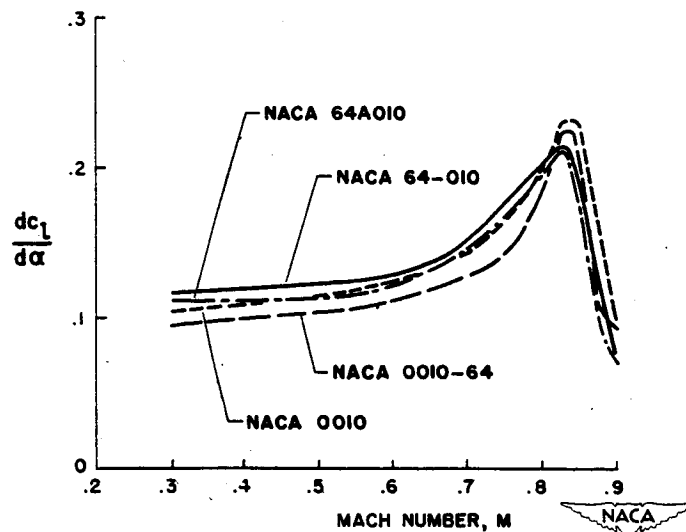


Figure 16.— Effect of thickness distribution on the variation of lift-curve slope with Mach number.

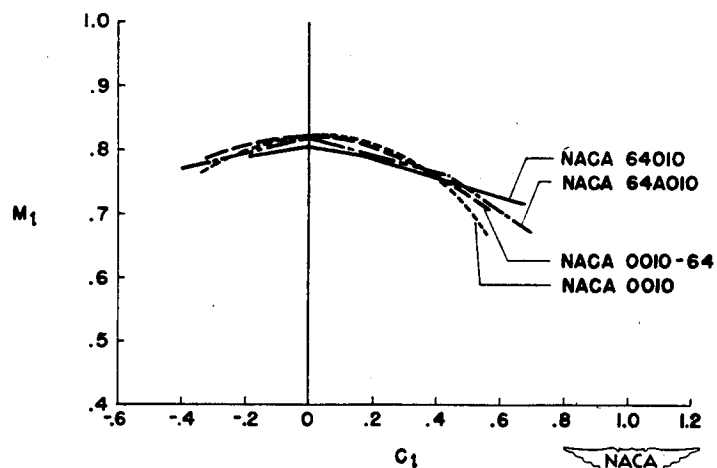


Figure 17.— Effect of thickness distribution on the variation of lift-divergence Mach number with lift coefficient.

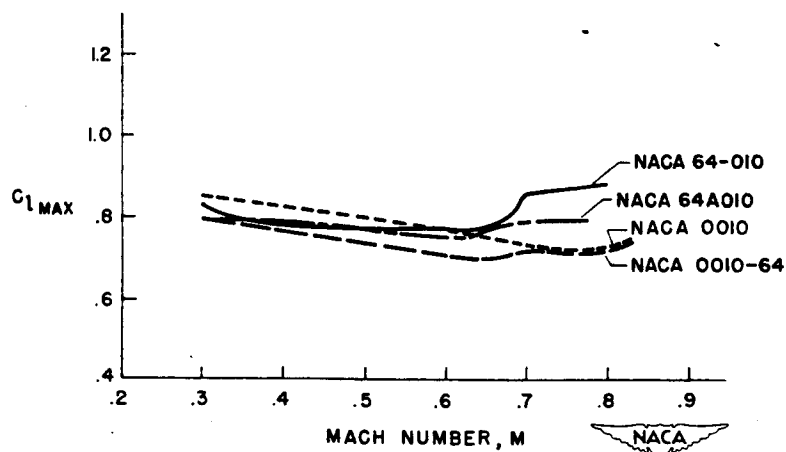


Figure 18.— Effect of thickness distribution on the variation of maximum lift coefficient with Mach number.

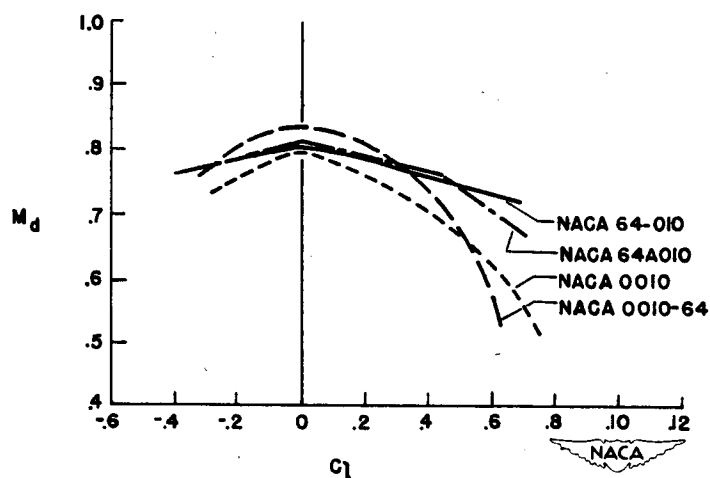


Figure 19.— Effect of thickness distribution on the variation of drag-divergence Mach number with lift coefficient.

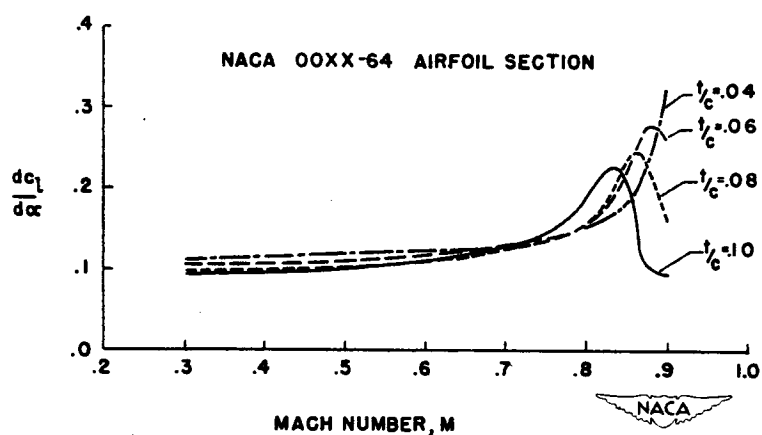


Figure 20.— Effect of thickness-chord ratio on the variation of lift-curve slope with Mach number.

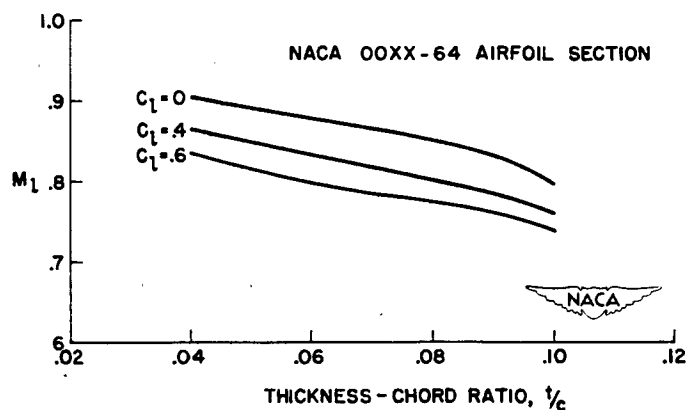


Figure 21.— Effect of thickness-chord ratio on lift-divergence Mach number.

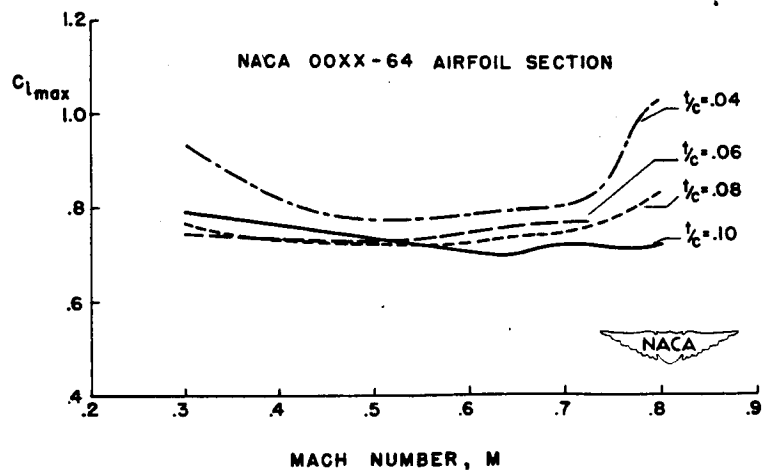


Figure 22.— Effect of thickness-chord ratio on the variation of maximum lift coefficient with Mach number.

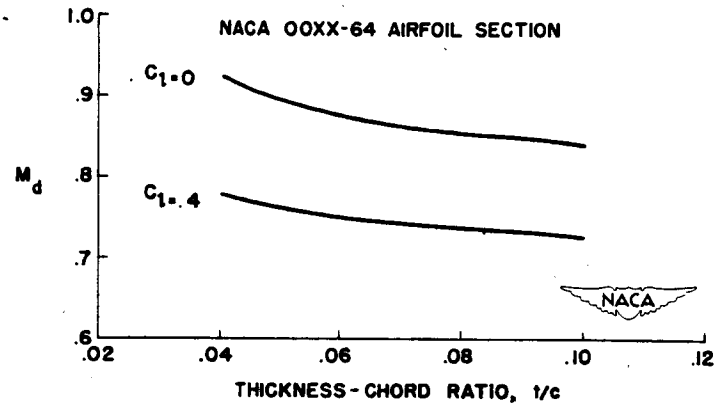


Figure 23.— Effect of thickness-chord ratio on drag-divergence Mach number.

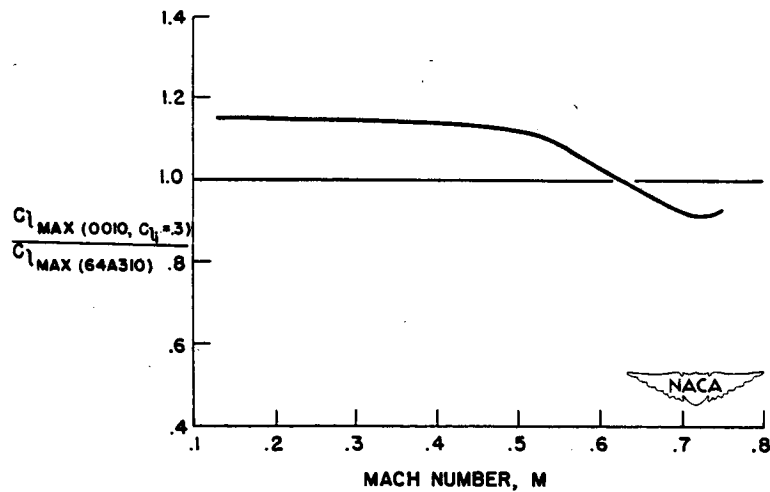


Figure 24.— Comparison of the variation of maximum lift coefficient with Mach number for two equally cambered NACA airfoils differing only in thickness distribution.

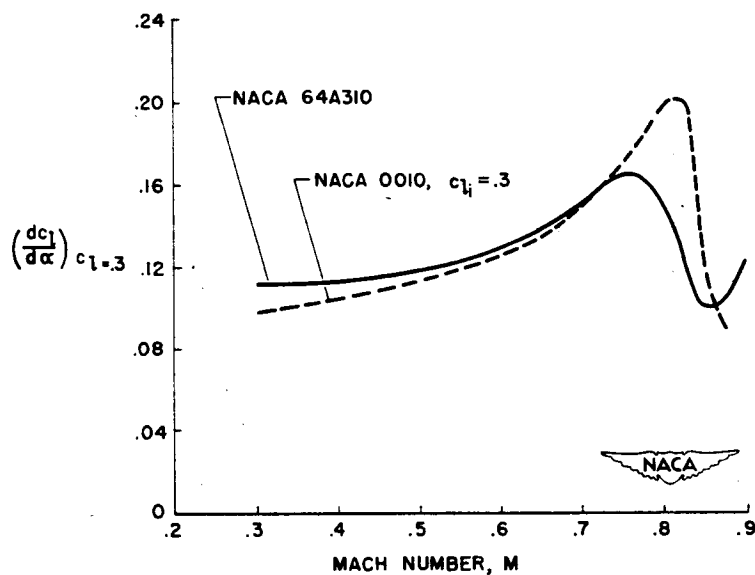


Figure 25.— The variation of lift-curve slope with Mach number for two equally cambered NACA airfoils differing only in thickness distribution.

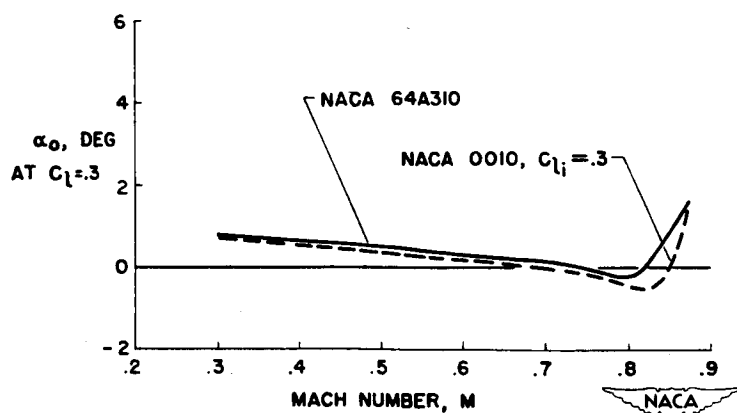


Figure 26.— The variation with Mach number of the angle of attack for the design lift coefficient for two equally cambered NACA airfoils differing only in thickness distribution.



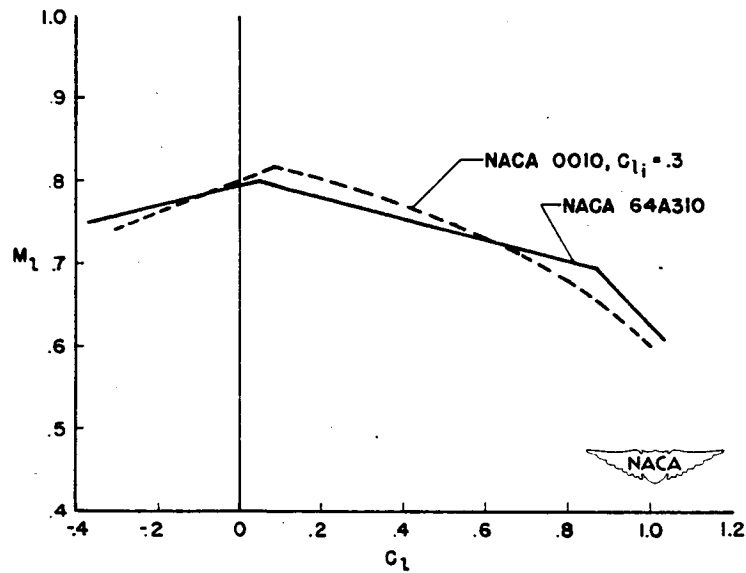


Figure 27.— The variation of lift-divergence Mach number with lift coefficient for two equally cambered NACA airfoils differing only in thickness distribution.

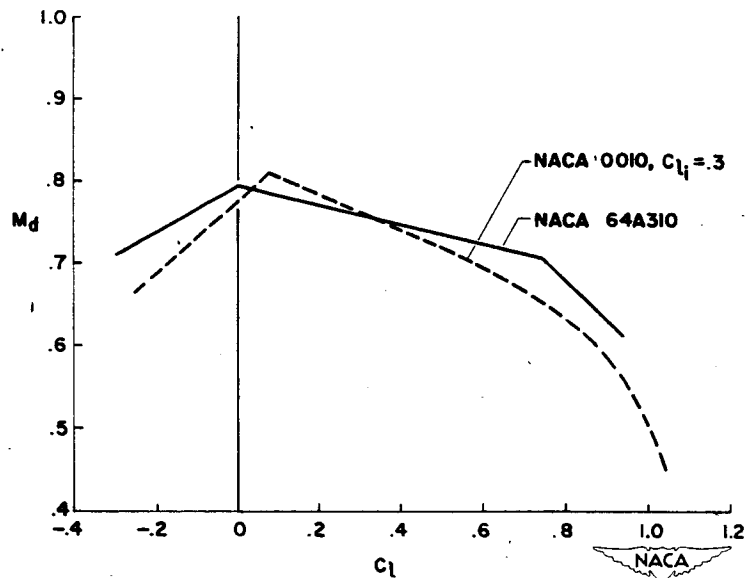


Figure 28.— The variation of drag-divergence Mach number with lift coefficient for two equally cambered NACA airfoils differing only in thickness distribution.